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THE SHOCK-WAVE NOISE PROBLEM OF SUPERSONIC AIRCRAFT IN STEADY FLIGHT

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SUMMARY

Data are presented which provide an insight into the nature of the shock-wave noise problem, the significant variables involved, and the manner in which airplane operation may be affected. Flight-test data are also given, and a comparison with the available theory is made. An attempt is also made to correlate the subjective reactions of observers and some associated physical phenomena with the pressure amplitudes during full-scale flight.

It is indicated that for the proposed supersonic transport airplanes of the future, booms on the ground will most probably be experienced during the major portion of the flight plan. The boom pressures will be most severe during the climb and descent phases of the flight plan. During the cruise phase of the flight, the boom pressures are of much lesser intensity but are spread laterally for many miles. The manner in which the airplane is operated appears to be significant; for example, the boom pressures during the climb, cruise, and descent phases can be minimized by operating the airplane at its maximum altitude consistent with its performance capabilities.

INTRODUCTION

In order to operate supersonic aircraft, it will be necessary for the commercial operator to recognize not only the noise problems associated with power plants and the aerodynamic boundary layer, but also the problem of the so-called sonic boom. Accordingly, data are presented which provide some insight into the nature of this shock-wave noise problem, the significant variables involved, and the manner in which airplane operation may be affected. An attempt is also made to correlate the subjective reactions of observers and some associated physical phenomena with the pressure amplitudes during full-scale flight.

SYMBOLS

K_1 ground-reflection constant

K_2 body-shape constant

l body length

l/d body fineness ratio

M airplane Mach number

p_a ambient pressure at altitude

p_0 ambient pressure at ground level

Δp pressure rise across shock wave

x distance to maximum thickness along body length

y distance normal to flight path

Subscript:

MAX maximum

NATURE OF THE PROBLEM

As an aid in understanding the nature of the problem, a schlieren photograph of a small airplane model is shown in figure 1. This is a profile view taken at a Mach number of 2.0 during wind-tunnel tests. The figure shows that there are strong shock waves attached to the bow and tail of the body, with additional shock waves emitting from other airplane components such as the wing. As these shock waves extend outward, they coalesce into the bow and tail waves which are gradually spreading apart or diverging. This divergence results from the difference in propagation velocities of the bow and tail waves which are, respectively, higher and lower than ordinary sonic velocity. The main reason for these differential velocities of propagation is, as indicated by reference 1, due to the longitudinal particle velocity or streaming velocity of the air which is always associated with shock waves. This same general shock-wave pattern is observed, whether lift is present or not, for bodies of various sizes and shapes and for full-scale airplanes.

In figure 2 is shown a schematic diagram for the test condition of figure 1. A slice through the wave pattern, as indicated by the horizontal line, would yield the pressure distribution shown by the heavy line. At the bow wave a compression occurs in which the local pressure rises to a value Δp above atmospheric pressure. Then a slow expansion occurs until some value below atmospheric pressure is reached, and then there is a sudden recompression at the tail wave. Again it is seen that the bow and tail waves are diverging, and, if the airplane were at an altitude of 40,000 feet, the time between these peaks would be about 0.2 second, which corresponds to a distance of the order of two to three times the length of the airplane.

If these waves were sweeping past an observer on the ground, the ear would respond as shown schematically in the sketch at the bottom of the figure. Since the ear is sensitive only to sudden changes in pressure, it would respond to the steep part of the wave and not to the portion which is changing slowly. If the time interval between these two rapid compressions is small, as for a bullet, the ear would not be able to discriminate between them and they would seem as one explosive sound. If the time interval were on the order of 0.10 second or greater, as in the case of the airplane, the ear would probably detect two booms.

SIGNIFICANT VARIABLES

There are many significant variables involved in the problem of the boom. These variables include those associated with the shock-wave generation in addition to those associated with the wave propagation through the atmosphere. Some of these variables are involved in the following equation, taken from references 1 to 3, which is used to predict the intensity of the boom:

$$\Delta p = K_1 K_2 \left(\frac{\sqrt{p_a p_0}}{y^{3/4}}\right)(M^2 - 1)^{1/8}\left(\frac{1}{l/d}\right) l^{3/4} \tag{1}$$

A discussion of the relative significance of the terms in the equation follows, but before the discussion, it is necessary to consider the equivalent-body concept. The pressure field at large distances from an airplane can be approximated as that from a body of revolution having the same length and maximum cross-sectional area. As in the "transonic area rule," this area includes not only sections from the fuselage but also from the wing, nacelles, and so forth.

In equation (1) the constant K_1, which is a reflectivity factor varying between 1.0 and 2.0, depends upon the ground surface, and for the particular tests considered herein it averaged between 1.7 and 1.9. The constant K_2 depends upon the equivalent body shape, and a theoretical variation of the constant is shown in figure 3.

The data of figure 3 are for three bodies of revolution having their maximum areas occurring at about 0.3 and 0.5 body length and at the tail of the body. It can be seen that for this extreme variation in the location of the maximum thickness, the constant K_2 varies from about 0.55 to about 0.80. In addition, wind-tunnel tests have indicated that bodies having the same longitudinal area development but widely different shapes give about the same value of Δp at large distances from the body or in the far field.

Figure 4 shows the theoretical variation of Δp with the remaining variables in the equation: altitude, Mach number, fineness ratio, and body length. Initial conditions indicated by the arrows are $l/d = 14.0$, $l = 190$, $M = 2.0$, and an altitude of 40,000 feet.

It should be noted that altitude has a twofold effect; namely, that of distance and of ambient pressure. It is obvious from figure 4 that the shock pressure rise decreases very rapidly with altitude. The pressure increases as Mach number increases but at a very slow rate above $M = 1.1$. Increasing body fineness ratio is beneficial in that the pressure varies inversely with the fineness ratio. Increasing the body length while maintaining the same fineness ratio is detrimental.

Equation (1) has been successfully used by many investigators to predict the pressures in the near field, as for the case of close passes of airplanes, where the distances involved are relatively small. (See refs. 4 and 5.) In this case the variables accounted for by the equation appear to be of primary importance. Equation (1) accounts for thickness only and does not include any effects of lift. The effects of lift in the downward direction are believed to be in phase with, and should add to, the effects of thickness and result in increased shock strength. The contribution of the lift is believed to be small at moderate altitudes but may become of greater importance for high-altitude operations. (See ref. 6.) In the present discussion an attempt is made to extend the use of this equation to the case of the observer on the ground, or in the far field, wherein the distances involved are relatively large. For this case, variables such as wind direction, wind velocity and temperature gradients, airplane flight path, and atmospheric losses may be of importance.

As an introduction to the far-field noise problem, or the exposure of people and structures on the ground to the shock front, it is helpful to review briefly the phenomena of shock-wave propagation. In general, a shock wave will not extend to the ground (and, consequently, no boom will be heard) unless the airplane local free-stream Mach number is greater than unity and the airplane velocity at altitude is greater than the velocity of sound at the ground. Thus, depending on the effects of existing temperature and wind conditions on shock-wave propagation, a boom may or may not be heard when the airplane is operated at supersonic Mach numbers. By assuming the conditions of a standard atmosphere, these propagation phenomena are illustrated in figure 5 for two steady-flight conditions where, for simplicity, only the bow waves are considered.

At a Mach number of 1.1 the bow wave does not extend all the way to the ground. If the temperature were constant at all points between the airplane and the ground, the bow wave would take the position of the dashed line and would intersect the ground. There is, however, a temperature gradient present; the ambient temperature at ground level is higher than at altitude. This temperature gradient affects the shape of the wave because the lower extremities propagate faster than the upper extremities and, thus, result in a "bending forward" of the wave as shown. This temperature effect is beneficial since, in some cases at low supersonic Mach numbers, it causes the wave to miss the ground completely. In the second case, where the local airplane Mach number is 2.0, the speed of the airplane exceeds the speed of sound at ground level and the wave front reaches the ground despite its curving because of the temperature gradient. Wind gradients have similar effects on the wave propagation and may either increase or decrease the curvature of the wave.

In connection with the pressures on the ground, therefore, it is recognized that there is a flight regime at low Mach numbers where the shock wave may not reach the ground. This condition is significant with regard to operations at low supersonic speeds in acceleration and deceleration near airports. However, in the steady-flight condition of higher supersonic Mach numbers, it is apparent that the shock wave will reach the ground, and this condition is the one primarily discussed herein.

OPERATIONAL FACTORS

The data of figure 6 show the pressure changes associated with the booms experienced near the flight track from some practical steady-flight operations of a supersonic airplane at various altitudes. The theoretical curve shown was calculated for a McDonnell F-101J airplane by using equation (1) and assuming $K_1 = 2.0$, $K_2 = 0.645$, $M = 1.3$, and $l/d = 7.7$ where $l = 67$ feet. The four experimental data points shown were recently measured for the airplane in the Mach number range from 1.25 to 1.4 and an altitude range of 25,000 to 45,000 feet. The circled data points were taken under different atmospheric conditions than were those designated by the square symbols. Examination of atmospheric-sounding data for the circled data points indicated that there were moderate headwinds of from 0 to 30 feet per second at altitude. Atmospheric-sounding data associated with the square symbols indicated a very similar temperature gradient, closely simulating that of the standard atmosphere, but a tailwind of from 60 to 100 feet per second existed at altitude. The data seem to fall into two groups, both of which show the same relative decrease with altitude as the theory predicts. It will be noted that the data fall on each side of the calculated curve for these two widely varying atmospheric conditions. Although not enough data are available for definite conclusions to be drawn, it does appear that equation (1) should be used with caution to predict pressures in the far field in cases of extreme variations in atmospheric conditions.

So far, only the observer on or near the flight track has been considered. From practical considerations an investigation of the extent to which the boom spreads outward from the track is also of interest. Some insight into this phenomenon is given in figure 7 in which both calculated and experimental data are again given for the F-101J airplane at an altitude of 35,000 feet traveling at a Mach number of 1.3. In this figure the pressures are shown as a function of lateral distance from the track in miles. The theoretical curve is given by equation (1), where y represents the slant distance from the airplane to the observer station. This calculation of equation (1) indicates a maximum pressure along the flight track, a decreasing pressure with increasing lateral distance, and a sudden "cutoff" due to refraction effects. The refraction effects arise from the previously discussed temperature gradients. (See fig. 5.) As indicated schematically in the sketch in figure 7, the ray paths emitting from the airplane are turned upwards as they approach the ground. The experimental data confirm, in general, the trends predicted by equation (1) and, in particular, the extent to which the boom spreads laterally. In that the terrain between observer stations was fairly flat and the surface winds quite low, it is believed that the effects of terrain and surface winds on the results indicated were minor.

Equation (1) has also been used to predict the pressure along the flight track of a possible future Mach number 3.0 transport airplane with fineness ratio l/d of 12.8 and a fuselage length l of 208 feet. These results are presented, along with similar calculations for the F-101J airplane at a Mach number of 1.3, for comparison in figure 8. Perfect reflection was again assumed in both cases, and a value of K_2 of 0.61 was taken for the Mach 3.0 transport airplane. At a given altitude, the difference in the calculated pressures is mainly due to the size and shape of the two airplanes; the other factors are secondary. (See fig. 4.) It is apparent that the pressures associated with the future transport airplane are higher, but it should be remembered that this airplane will cruise at much higher altitudes.

The following table, which is based on the material of reference 7 and the present tests, attaches some significance to the order of magnitude of pressures to be expected.

Shock-noise phenomena			
Δp, lb/ft^2	Δp, decibels	Resulting physiological reaction	Associated physical phenomena
0.1 to 0.3	108 to 118	Not objectionable	Barely audible explosion
0.3 to 1.0	118 to 128	Tolerable	Distant explosion or thunder
1.0 to 3.0	128 to 138	Objectionable	Close-range thunder, some window damage
3.0 to 10.0	138 to 148		Damage to large plate-glass windows
10.0 to 30.0	148 to 158		Definite damage to small barracks-type windows

This table presents some shock-noise phenomena for various pressures along with the equivalent decibel values. Also indicated in the table are some observations by people who have experienced this type of noise along with some well-known physical phenomena that occur at the same pressure values. For the particular tests of this investigation where the ground pressures did not exceed 1.0 pound per square foot, the observers did not consider the booms objectionable and likened them to a distant thunder or explosion. For pressures exceeding 1.0 pound per square foot the observers considered the boom objectionable. For higher pressures of from 3.0 pounds per square foot to 30.0 pounds per square foot it was indicated in reference 7 that damage to large plate-glass windows and smaller barracks-type windows would occur.

During the flight at 25,000 feet altitude damage to a large plate-glass store window was correlated in time with the overhead passage of the airplane. This damage consisted of a nearly horizontal crack across the upper portion of a window having a vertical dimension of 128 inches and a horizontal dimension of 90 inches. Although pressures were not measured at the site of the damage, the ground pressures along the track of the airplane were estimated to be about 2.0 pounds per square foot. It should be noted that a steady loading of 2.0 pounds per square foot corresponds to a 28 mile-per-hour wind. Further, glass manufacturer's charts show that a window of this size, properly mounted, should be able to sustain a static load of about eight times this value. Window failure at the estimated ground pressure of 2.0 pounds per square foot therefore cannot be explained directly, but may have been due to installation stresses, inadequate support of the vertical edges, multiplication of pressures through shock reflection from ground and buildings, or multiplication of stresses due to dynamic response of the windows. However, in view of the fact that similar windows on either side of the cracked window did not break, it appears that the pressures incurred in this test were near the magnitude where damage might begin to occur for commercially installed plate-glass windows.

With an appreciation established for the mechanism of generation of the boom, for the operational and atmospheric factors that affect it, and for the associated physical phenomena, an examination was made of how the shock-wave noise problem may influence future airplane flight plans.

Proposed altitude profiles are shown in figures 9 and 10 for the previously discussed Mach 3.0 transport airplane on a cross-country flight, along with an indication of the intensity and lateral speed of the boom pressures. The nominal flight plan of figure 9 can be associated with optimum performance and the plan of figure 10 with having taken into consideration the "boom" problems. These data might be typical for supersonic transports, assuming conventional airframe and engine characteristics. In both plans the same amount of fuel is consumed, but in the optimum-performance plan the distance is covered in about 13 minutes less time. The only difference between the two plans is in the climb and descent phases, as shown. In the optimum-performance plan the climb and descent are made at maximum allowable indicated airspeed, which results in high supersonic speeds at low altitudes and, therefore, produces pressures of about 5 pounds per square foot along the flight track. During the cruise portion of the flight plan, which begins about 300 miles from take-off, the pressures are of much lesser intensity (0.5 pound per square foot) but extend laterally about 60 miles.

In an attempt to minimize the pressures during the critical phases of climb and descent, the alternate flight plan of figure 10 which results in the same fuel consumption but requires a longer flight time has been proposed. In this plan the climb and descent phases are made at subsonic speeds to some intermediate altitude of perhaps 35,000 feet, at which time the airplane is about 80 miles from the point of take-off, and then the Mach number is increased at constant altitude to about 2.0. A supersonic climb is then made to the cruise altitude of 70,000 feet, and the airplane has attained a distance of about 400 miles from the point of take-off. For this alternate flight plan, pressures of about 2.5 pounds per square foot are experienced during the climb and descent phases, as compared with 5.0 pounds per square foot for the previous plan of figure 9; the pressures during the cruise portion being the same. It is obvious from the results that the boom will be experienced for a major portion of the flight plan and will be most severe during the climb and descent phases. The acceleration and deceleration from supersonic speeds should be accomplished at as high an altitude as possible in order to minimize the boom pressures on the ground.

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CONCLUDING REMARKS

It has been pointed out that for the proposed supersonic transport airplanes of the future, booms on the ground will most probably be experienced during the major portion of the flight plan. The boom pressures will be most severe during the climb and descent phases of the flight plan. During the cruise phase of the flight, the boom pressures are of much lesser intensity but are spread laterally for many miles. The manner in which the airplane is operated appears to be significant; for example, the boom pressures during the climb, cruise, and descent phases can be minimized by operating the airplane at its maximum altitude consistent with its performance capabilities.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., November 6, 1958.

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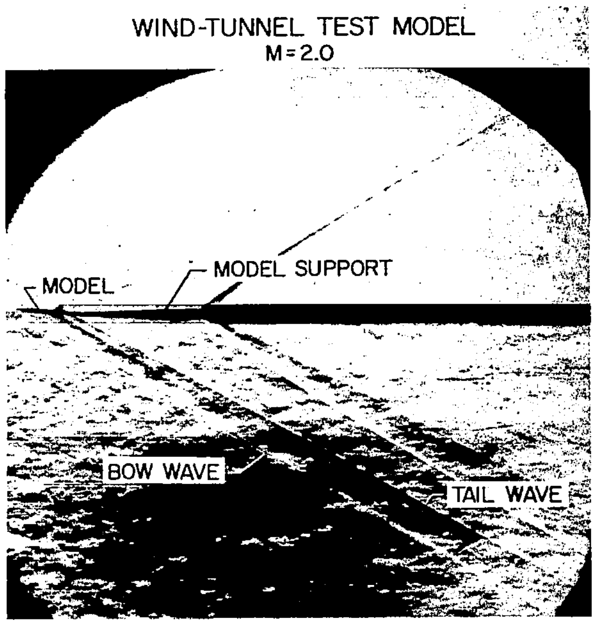


Figure 1

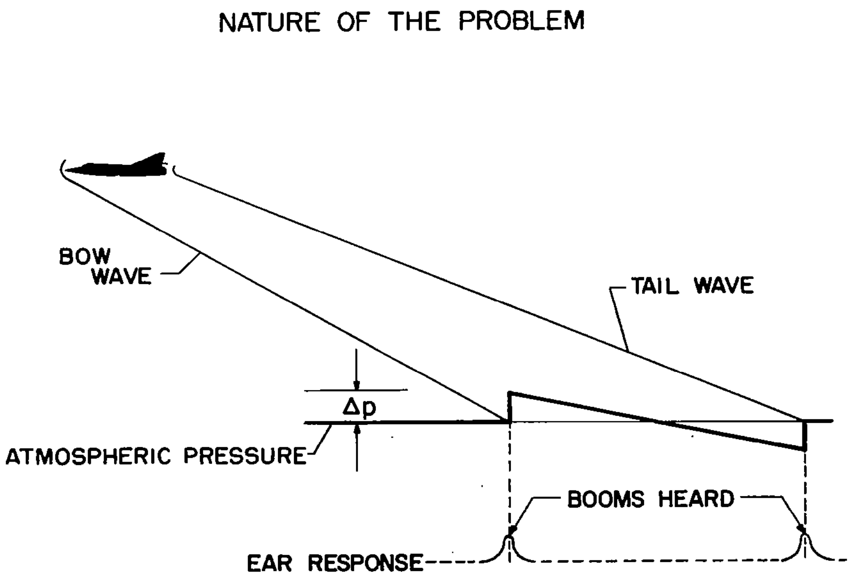


Figure 2

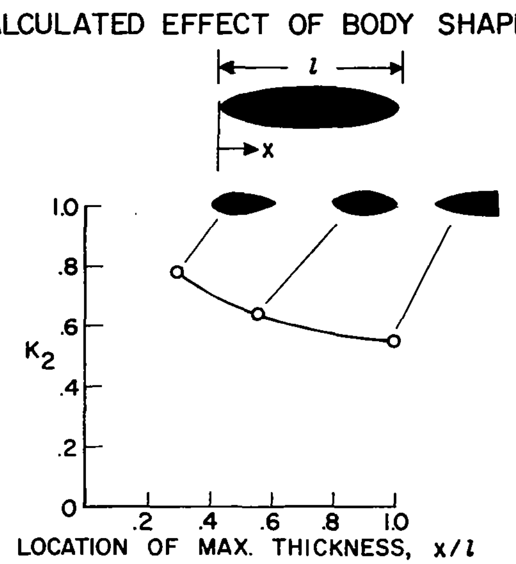


Figure 3

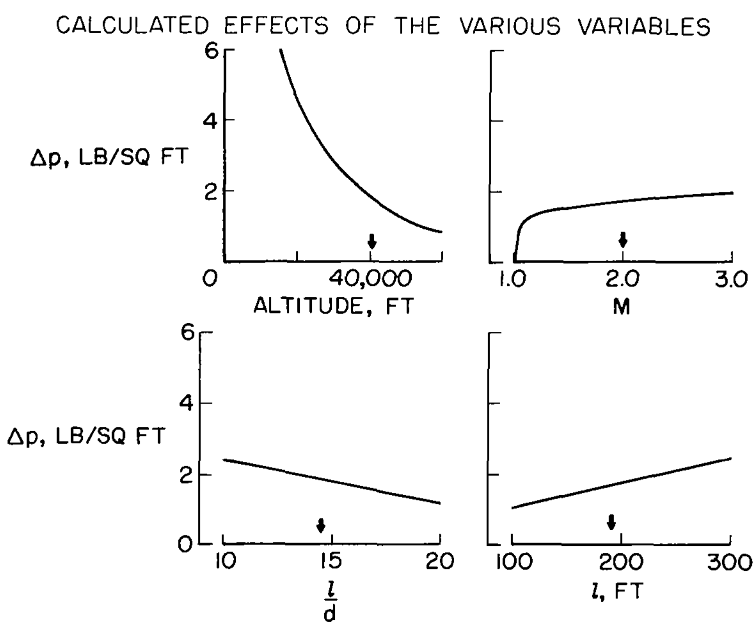


Figure 4

EFFECT OF TEMPERATURE GRADIENT ON
BOW-WAVE PROPAGATION

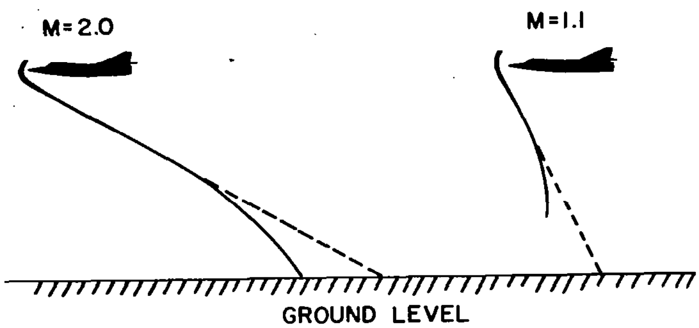
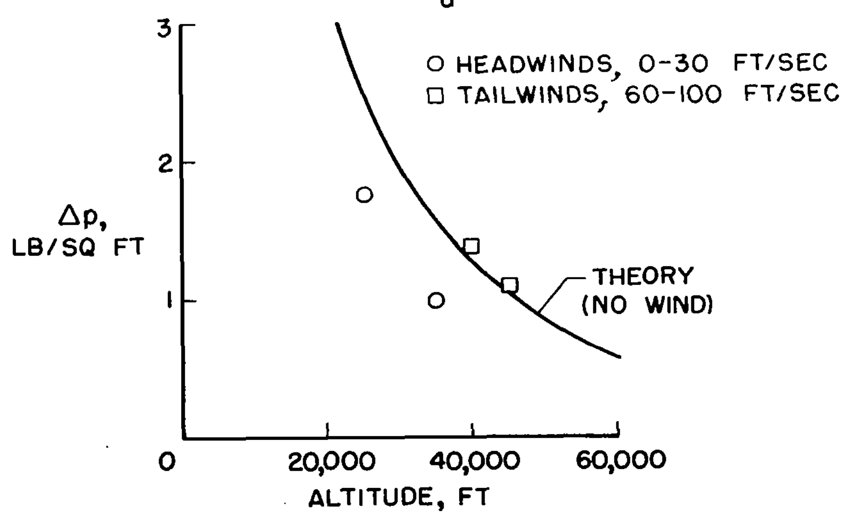


Figure 5

PRESSURES ON GROUND ALONG TRACK

$M = 1.3; \ \frac{l}{d} = 7.7$



F-101J
$K_1 = 2.0$
$K_2 = 0.645$
$M = 1.3$

Figure 6

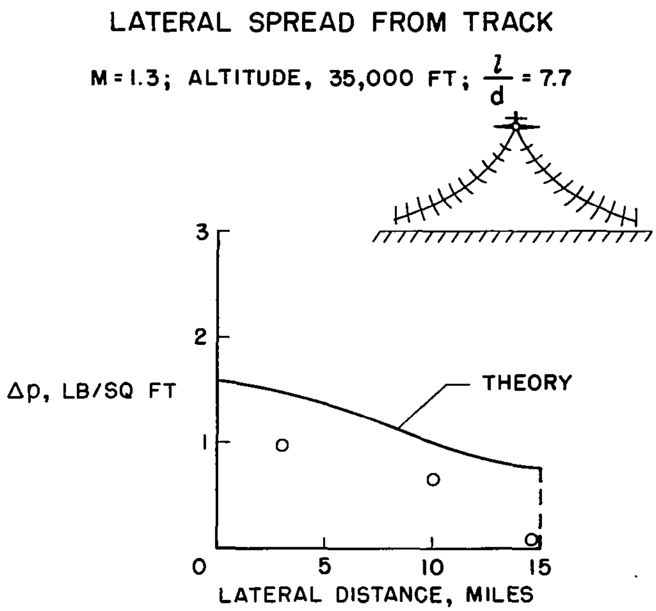


Figure 7

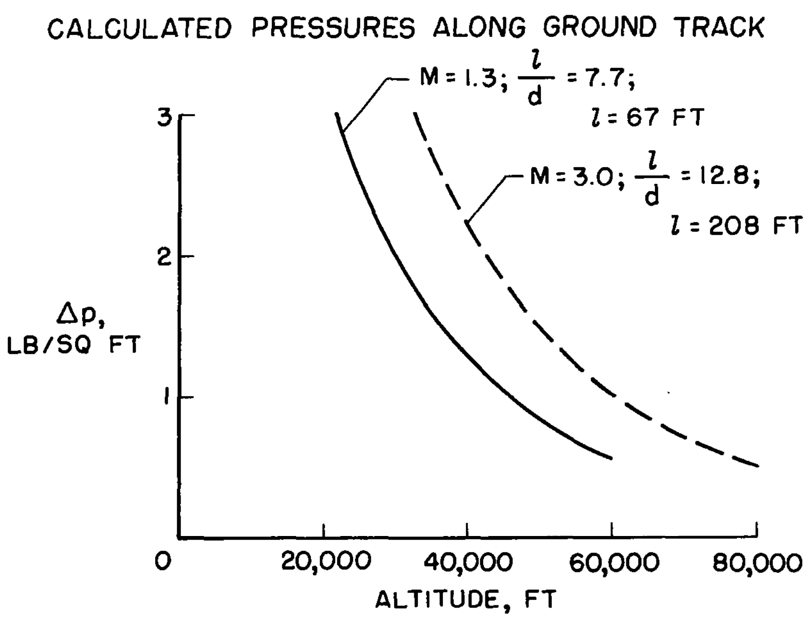


Figure 8

CROSS-COUNTRY FLIGHT PLAN

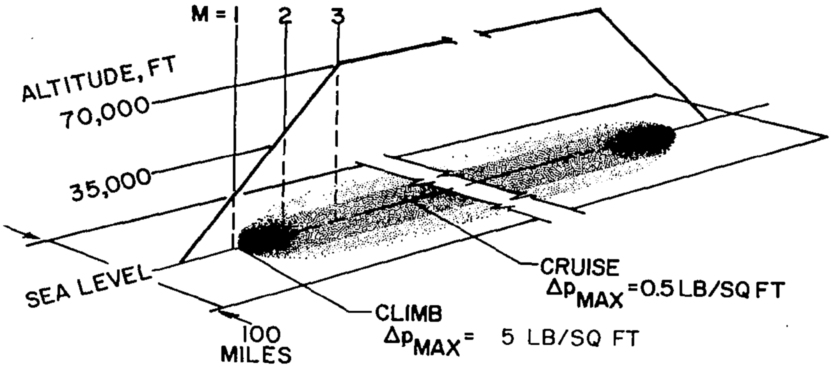


Figure 9

CROSS-COUNTRY FLIGHT PLAN TO REDUCE NOISE

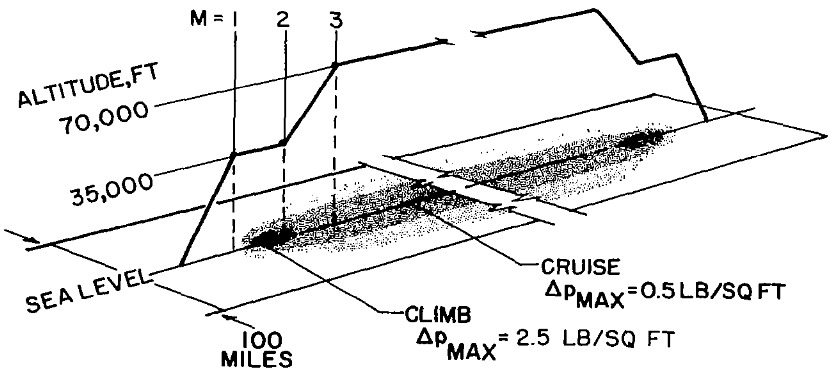


Figure 10

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